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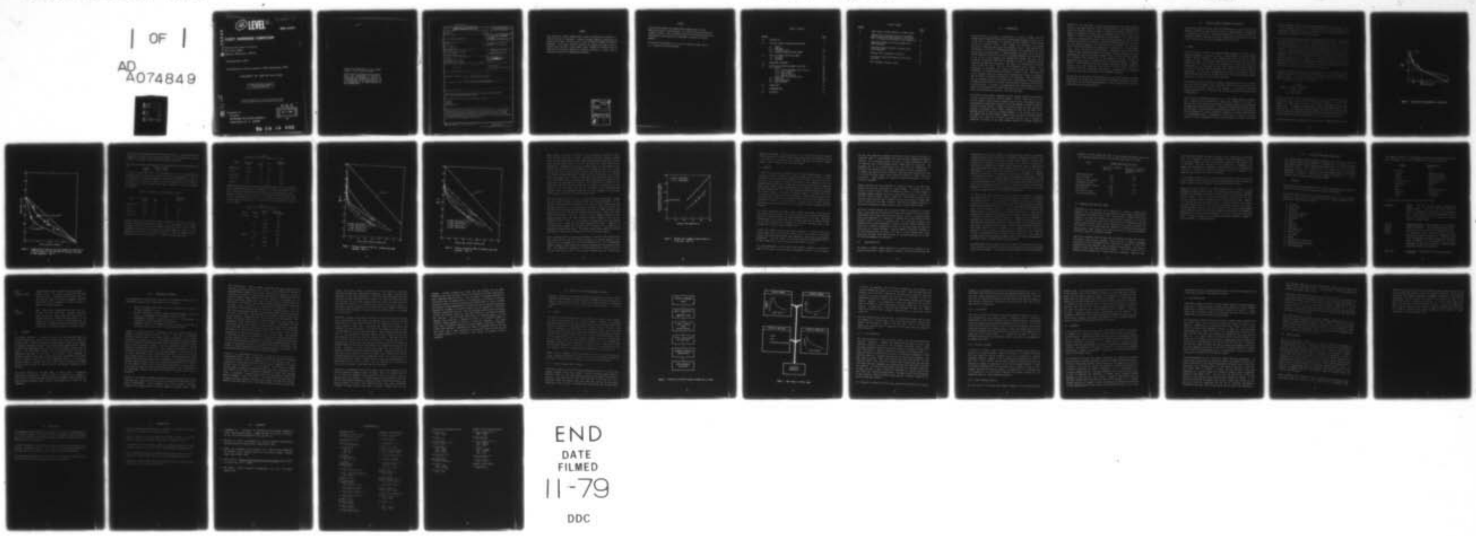

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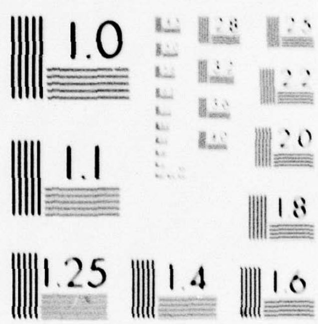

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FLEET HARDNESS VARIATION

Boeing Aerospace Company
P.O. Box 3999
Seattle, Washington 98124

29 September 1978

Final Report for Period January 1978—September 1978

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The variation of nuclear hardness in a fleet of aircraft is considered and the potential hardness degradation mechanisms identified. These include cracks, corrosion, paint deterioration, and aging of equipment. Methods are developed whereby the variation in hardness in a fleet could be estimated for each of these degradation mechanisms.

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SUMMARY

The variation of nuclear hardness due to blast, overpressure and thermal in a fleet of aircraft is considered. Potential hardness degradation mechanisms are identified as cracks, corrosion, paint or skin reflectivity changes and aging of equipment. The critical aircraft structure and systems are identified for a nuclear burst, and their failure modes listed. Methods are developed based on observation and analysis whereby the hardness variation in a fleet could be estimated for each of the degradation mechanisms.

PREFACE

This work was funded by the Defense Nuclear Agency under contract DNA001-78-C-0138 as a continuing part of a study of mission completion of strategic aircraft. Capt. M. Rafferty (USAF) was the Contracting Officer's Representative and we gratefully acknowledge his support and helpful comments.

The work was performed by Mr. E. N. York as Technical Leader, and Dr. S. L. Strack was the Program Manager.

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1.0 INTRODUCTION

Any major fleet of aircraft is continually in a state of change, with the changes being introduced intermittently and spreading through a portion, or all of the fleet. Some changes are introduced as block changes or modifications during the production phase. Other changes are introduced by equipment additions or modifications during the operational phase. Some changes occur unwittingly as a result of maintenance procedures or by substitution of parts during repair and replacement. A system designed and qualified to specified nuclear hardness levels will be varied in hardness knowingly or unintentionally by any or all of the equipment and configuration changes that occur during production, maintenance, modification programs, by aging of components and by structural fatigue during operation. The result is a distribution of hardness levels across the fleet. Some of the differences among aircraft can be determined, or estimated, from records of modifications and repairs. Some differences can be determined by inspections or special tests. The number and size of fatigue cracks for example can be determined by inspection procedures and the effect on structural strength can be calculated. Other differences, such as piecepart substitutions during repair and replacement and the vagaries of weather and aging lead to hardness variations which cannot be quantified from existing records. At present, the magnitude of these differences is not well known and the technology for assessing the impact of aircraft differences on the fleet mission-completion capability is not well developed.

To the best of the authors' knowledge, the only hardness controlled operational military systems have been missile systems. The B-1 bomber was the first aircraft system in which nuclear hardness specifications and hardness control procedures were implemented during the design, development, prototype production and qualification testing. A number of aircraft systems have been analyzed or assessed to determine their nuclear hardness but they are not under strict hardness control to retain the assessed hardness. A few aircraft such as the E3A (AWACS) and the E4B (Airborne Command Post) have partial hardness control programs, but since the airframes are taken directly from a commercial production line which is not under hardness control the airframe production

hardness is not controlled. On missile programs which are under hardness control it has been found that one of the key elements is that maintenance and operational personnel must be trained to recognize the hardness implications of defects or configuration changes that may otherwise appear quite trivial. The emphasis of normal missile maintenance and operations is for high launch reliability and operational "ready" status of mission equipment. This has to be augmented by deliberate attention to potential hardness degradation to assure that the expected hardness of the missile system is retained. The emphasis of normal aircraft maintenance and operations is primarily on air worthiness, flight safety, and mission system functions. Seemingly trivial configuration changes or parts substitution which could affect aircraft hardness may not be highlighted by existing maintenance records or by air crew in-flight records. This introduces some uncertainty in the adequacy of current records for determining the aircraft to aircraft variations in nuclear hardness. In general however, aircraft operation/training records, maintenance records, inspection records, test order compliance records and air crew reports provide a substantial base for determining the specific configuration of individual aircraft and from this, to determine likely hardness variations.

The data base for determining fleet hardness variations thus consists of normal configuration and flight time record keeping, the results of special one time inspections, and the experience/background of people concerned with aircraft nuclear hardness assessment programs.

2.0 POTENTIAL HARDNESS DEGRADATION MECHANISMS

This section discusses the basic physical processes that can cause structure and systems to degrade in load carrying capability. Where possible, the reduction in capability is quantified. Cracks and corrosion are discussed as they affect structural elements, and aging (or really, another form of corrosion) is related to skin reflectivity and avionics components and black boxes.

2.1 CRACKS

Aircraft structural integrity has generally been considered a function of its specified strength or ability to resist a limited number of carefully chosen ultimate loading conditions. This integrity has been verified by design, analysis and test. In most cases the tests are static tests to the limit load conditions with the ultimate load capability verified by analytical extrapolation. In a few cases one particular loading condition was chosen for an ultimate load test, and the structure was tested to this load or even to destruction. On this well instrumented test article the analytical stress predictions were verified at discrete points and failure mode(s) observed and repaired up to a major failure or end of the test.

MIL-STD-1530 A (Aircraft Structural Integrity Program) now calls for an ultimate load test to be the general rule, and proof load (or limit load) tests to be used only with specific approval in the contract. Even with this provision however only one article is tested to the ultimate load and no statistical base is established.

This MIL-STD also emphasizes the need for a total program (design, analysis, test, inspection and maintenance) to ensure the integrity of aircraft in service, due to the recognition of durability problems caused by crack formation and growth and corrosion, wear etc. A major and increasingly successful effort has been made to identify problem areas before they become critical by requiring fail safe design with suitable inspections intervals, or fatigue tolerant design in non-inspectable areas. Consequently a major effort in analysis test and design has been accomplished to ensure that structures, as built, will have

adequate strength, in spite of crack growth, between inspection intervals. That is, the problem solved is one of predicting when a crack may grow to an inspectable length while subject to a normal service load spectrum and retaining some specified (generally limit load) strength.

For the nuclear hardening assessment case, however, the problem is to determine the existing crack population in the aircraft fleet at any particular time, and determine the reduction in ultimate load capability due to these cracks. In this section we will evaluate the reduction in strength due to small cracks in materials. The reference works in this area generally relate only to residual strength of sample test panels rather than built up members and major structural components, but it will be assumed that a reduction in material strength will be directly proportional to a reduction of structural strength. It must be emphasized that this reduced strength with cracks applies to tension loads only. Compression and buckling critical members must be considered separately and their failure modes studied for any tension critical areas.

Feddersen (Reference 1) presents a unified correlation to the problem of center cracked panels that gives useful design information in residual strength. As shown in Figure 1 he plots the stress-crack length curve for ideal elastic fracture behaviour of a panel in plane stress, i.e.

$$K = S\sqrt{\pi c}$$

where K = fracture toughness
S = stress level
$2c$ = crack length

He then joins targets to this curve from the limit points, on the left from the yield stress (S_{ty}), and on the right from the panel width, W. The left hand tangent, for high stress and small crack length accommodates plastic zone effects, and is of immediate interest here. Test data has shown (Figure 2) that this composite curve represents material behavior very well, and can be applied to the K curve for threshold damage, where there is stable crack growth - more or less corresponding to the yield stress (S_{ty}) on the stress-strain curve.

It would appear that the appropriate limit point for a tangent from the K-curve

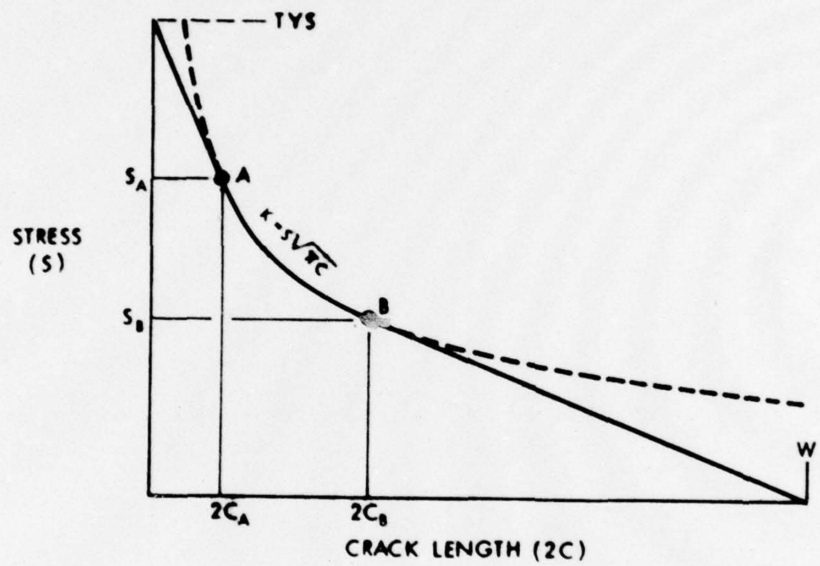

Figure 1 Ideal elastic fracture behavior in plane stress

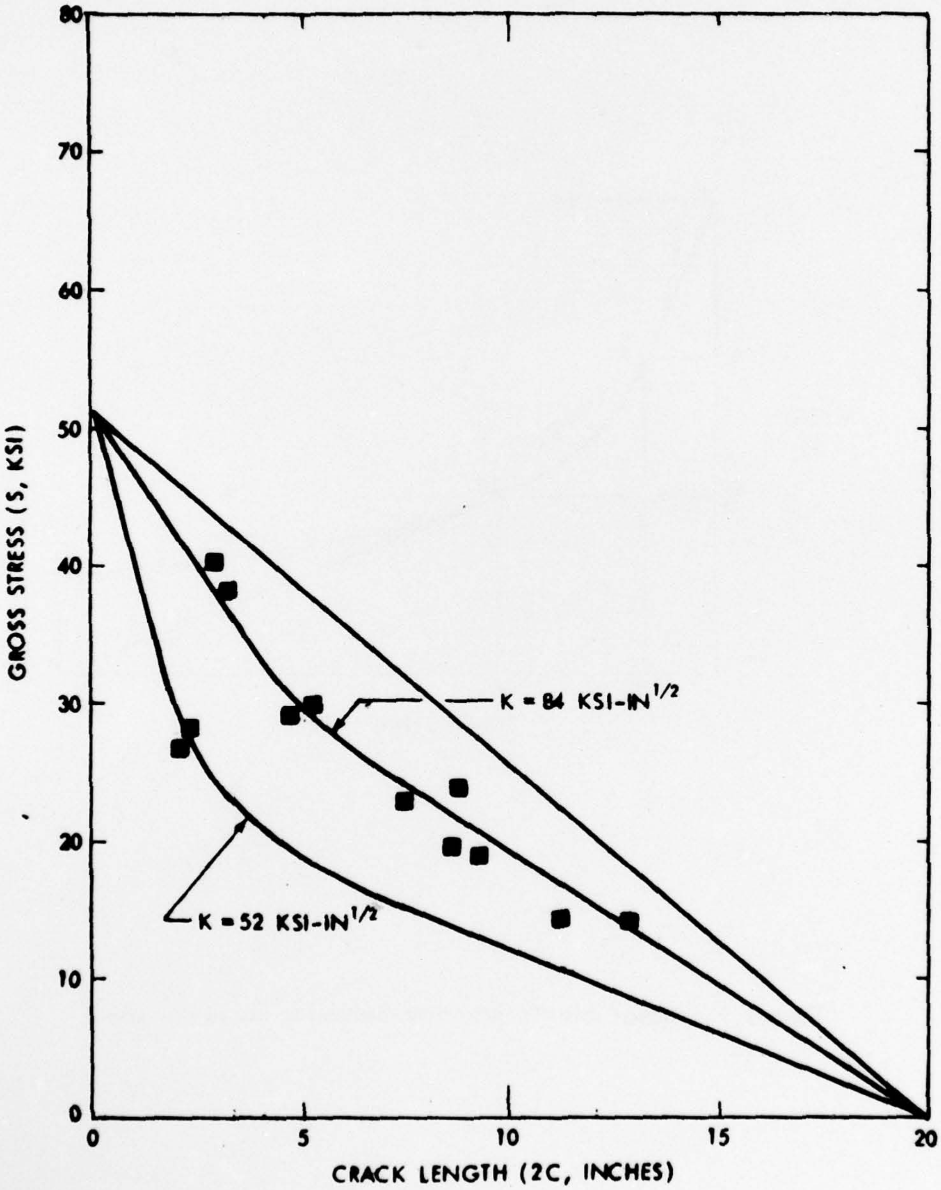

Figure 2 Computer plot of data and curves for threshold and criticality of 0.040 inch thick, 20 inch wide 2024-T3 aluminum alloy sheet at room temperature. (Ref. 1)

for critical crack growth would be the ultimate stress S_{tu}. Since design stress allowables are based on the yield stress however, it is convenient to use this quantity as a measure of the strength reduction of the panel.

The slope of the left hand target can be obtained with some simple manipulation as

$$\left(\frac{dS}{d2c}\right)_1 = -\frac{2\pi S_{ty}}{27}\left(\frac{S_{ty}}{K}\right)^2$$

(It must be noted that the K in this equation is the K corresponding to threshold fracture instability, and not the usual K corresponding to critical crack growth.) A convenient method of visualizing this reduction in strength is to convert this to a percent reduction in yield strength per inch of crack, by dividing by S_{ty}. This gives the results shown in Table 1 for the data presented by Feddersen.

Table 1 Reduction in Yield Strength (2c=0.1")

Alloy	Thickness (In.)	S_{ty} (ksi)	K	Percent Reduction
2024-T3 clad	0.04	51	52	[illegible]
7075-T6	0.09	75	43	7.1%
7075-T7351	1.00	61	33	8%
7075-T7351	0.25	61	41	5%

From this table it can be seen that cracks of only 0.1" can cause a significant reduction of the yield strength of a panel. If the same tangent fairing method is applied to the critical crack instability curves and taken to the ultimate strength on the y-axis we can obtain the results shown in Table 2. This suggests that for a 0.1" crack the load required to cause an unstable crack is reduced by 3 - 4% for 7075 alloys to 1 - 1.5% for 2219 and 2024 alloys.

TABLE 2
Reduction in Ultimate Strength (2c = 0.1")

Alloy	Thickness (In.)	S_{tu} (ksi)	K_c	Percent Reduction
2040T3 clad	0.04	68	84	1.5%
7075-T6	0.09	83	70	3.0%
7075-T7351	1.0	72	52	4.5%
7075-T735	0.25	72	52	4.5%
2219-T87	0.1	70	105	1.0%

Even greater reductions in residual strength were observed in an experimental program reported by McEvily et al (Reference 2) or sheet specimens of 7075 and 2024. Typical results based on their experiment are reproduced from their report as Figures 3 and 4. For the 7075-T6 specimens the reduction was about 40% for a 1% of width crack, and for the 2024 there was about a 30% reduction for the same 1% of width crack. Actual test data for the smallest cracks are given in Table 3.

TABLE 3
Results of Residual Strength Tests

Sample	Width	Cracked Width	Reduction in Strength
7075-T6	2 1/4"	1%	4%
		1.8%	22%
	12"	0.1%	20%
		0.1%	19%
	35"	0.8%	42%
2024-T3	2 1/4"	2%	24%
	12"	2.5%	29%
		1.1%	21%
	35"	0.2%	31%

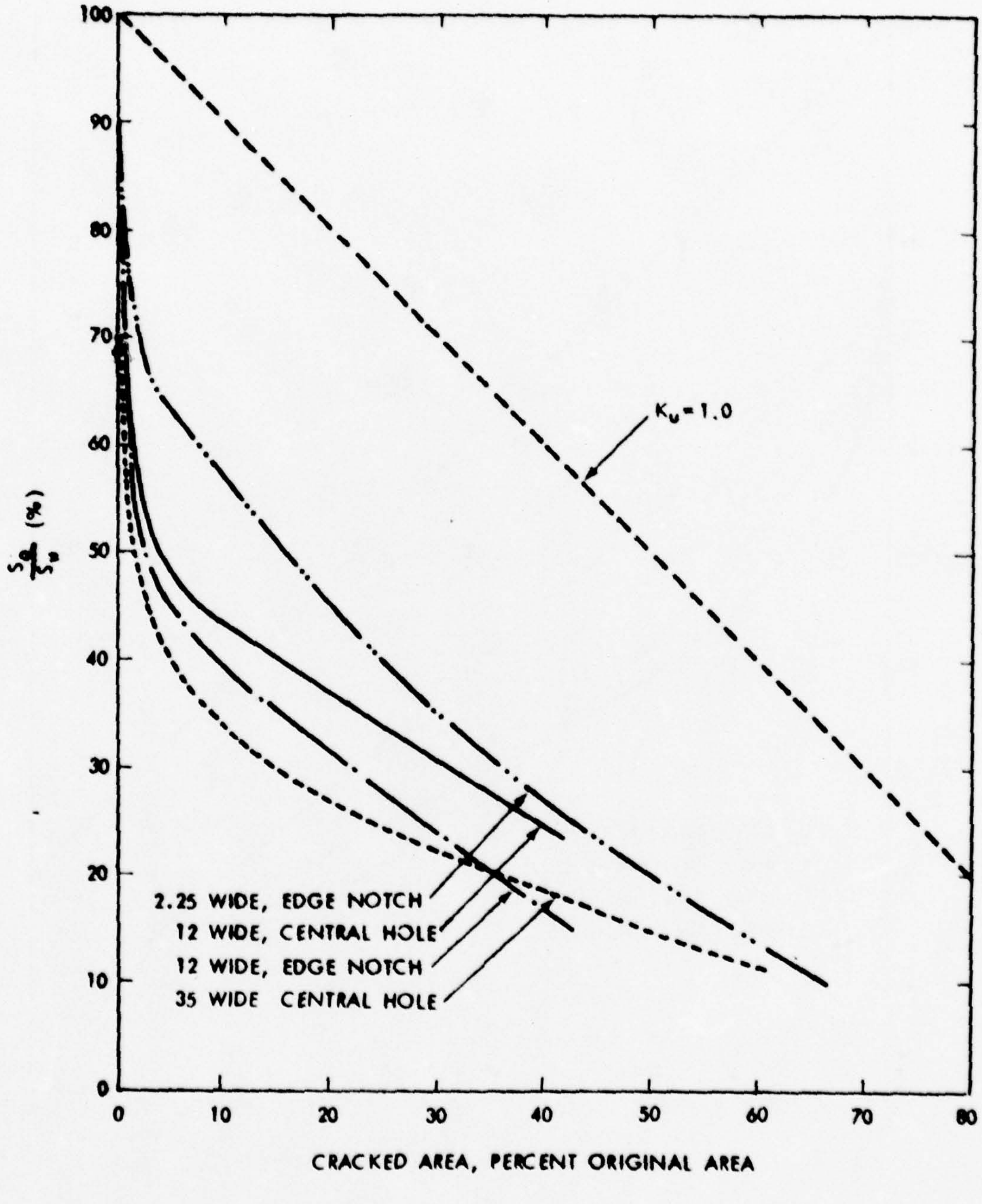

Figure 3 Predicted strengths of 7275-T6 aluminum alloy sheet specimens (Ref. 2)

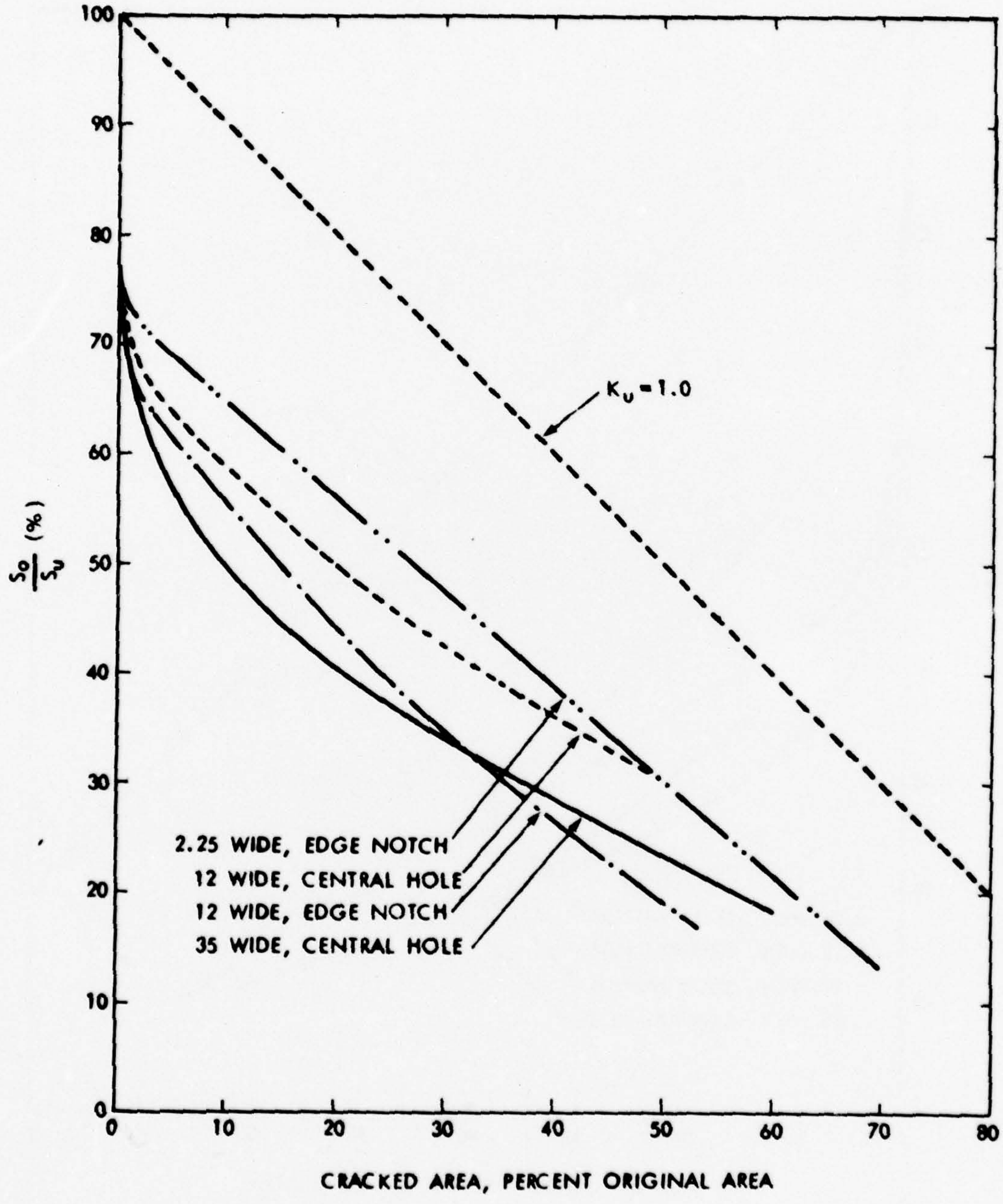

Figure 4 Predicted strengths of 2024 - T3 aluminum alloy sheet specimens. (Ref. 2)

These results, of course, all refer to tests performed on samples of sheet material rather than built up panels. Reference 3 describes residual strength tests made on 9 C-46 wings in 1956, where the wings were subjected to fatigue tests to grow fatigue cracks in critical areas and then tested statically to destruction. Apparently the critical spanwise station of the wings was well defined, for the cracks all were observed to originate in three areas at wing station 214 or 195. Figure 5 gives a plot of the reduction in static strenngth as a function of remaining area of tension material in the wing, and indicates about a 30% reduction in ultimate strength for a 10% reduction in tension area. For the particular failures involved, this 10% of tension area included one skin panel about 10" wide and two stiffners. The authors did not attempt to test wings with very small fatigue cracks but did observe "extrapolation of the results of the full scale tests to near the region of the undamaged wing indicates that a considerable reduction in static strength of the tension surface would result from a very small fatigue crack."

One final observation must be made. Figure 5 (copied from Reference 3) shows the limit load of the tension surface as 40% of its ultimate strength. This is because of the large margins of safety in the lower surface of the wing for the positive loading case - i.e. lower surface in tension. These large margins for the positive load case were dictated by the fact that the lower surface of the wing was critical in the negative loading (buckling) case, in which very small margins were present. That is, in order to have adequate strength for the negative design load, the lower surface had excess strength in the positive design load case. This is a very important consideration for any aircraft structure, for it means that we cannot automatically equate a reduction in ultimate tension strength of a structure with a reduction in its capability to withstand ultimate design loads. The actual design loading conditions must be considered in each case to determine the critical structural members.

Thus it appears that if a small crack existed in a section of the aircraft that were blast critical, the first ultimate loading in a nuclear encounter could cause a crack to propagate to its fail safe design limits. If in the subsequent dynamic response a load beyond limit load were imposed then the structure could fail. This reduction in hardness will be discussed further in section 5, but it must be noted that a crack in an aircraft structure does not automatically

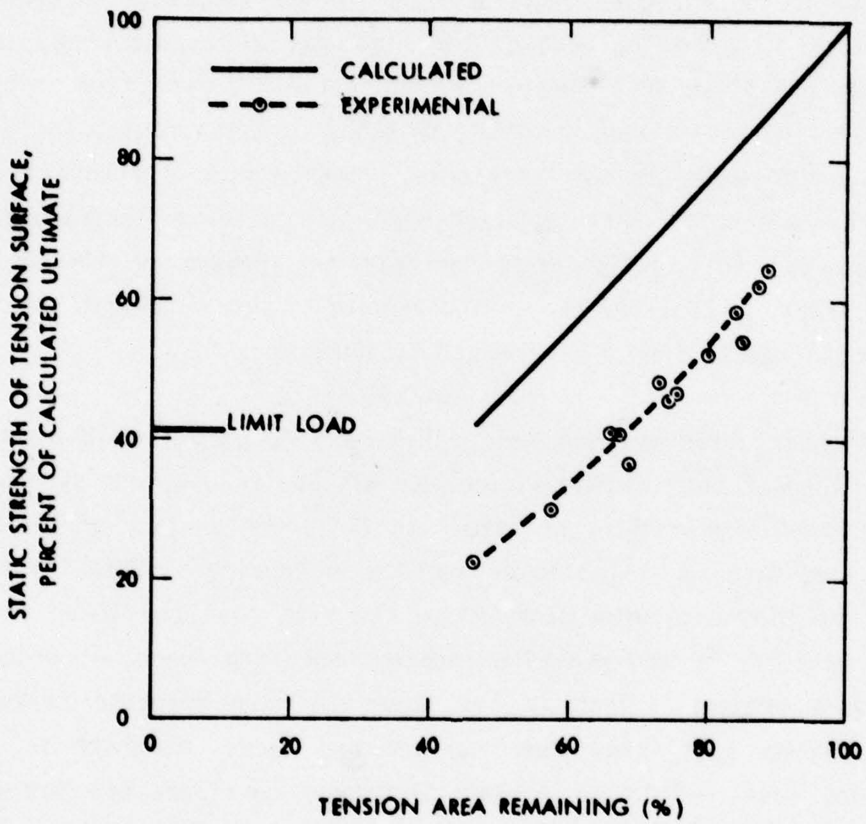

Figure 5 Residual static strength of tension surface of C-46 wing. (Ref. 3)

degrade the hardness. The crack must be in a critical section of the airplane that actually gets the design ultimate load, there must be zero margin of safety in this member and there must be a repeated load (due to structural dynamic response greater than the limit load for failure.

2.2 CORROSION

In spite of very effective commercial and military corrosion control programs that have evolved in response to corrosion incidents, water and other corrosive components can cause significant damage over a 20 year time span to aircraft structures and equipment. As with fatigue there is the recognition that this process cannot be completely stopped, but its effects can be greatly reduced by good process control during manufacturing and a continuing inspection program during the operational life. Air Force Regulation 400-44 "Corrosion Prevention and Control Program", MIL-STD-8F9 "Dissimilar Metals", MIL-SPEC MIL-F-7179E "Finishes and Coatings; Protection of Aerospace Systems Structures and Parts", all provide policy and direct actions which serve to minimize the probability of corrosion in service. However there are many cases in which adequate consideration has not been given to the preventive measures directed by these documents.

In aircraft structures, corrosion is most frequent and severe in lavatory and food service areas, integral fuel tanks, areas of the structure where moisture can co' ect, and areas where very high strength materials are used. Many high strength materials are used in the heat treatment condition for highest strength and are more sensitive and susceptible to corrosion than when given a heat treatment for a slightly lower strength level.

Different environments can cause very great differences in corrosion rates, and aircraft based near sea coasts (with traces of salt in the air) or in hot humid regions show marked deterioration rates compared to those based in more benign atmospheres. The dry climate at the DoD "boneyard" Davis-Monthan AFB in Arizona, preserves the aluminum airframes stored there very well.

In a study sponsored by the Air Force Flight Dynamics Laboratory, corrosion was identified as a major cause of electronic equipment failures. In one instance

the Air Force Materials Laboratory identified stress corrosion cracking in transistor leads in a ground-based radar which had been experiencing numerous failures. This cracking occurred as a result of contact between the leads of the transistor and a fibrous washer material which acted as a lamp wick. The equipment was in an air-conditioned area, and while the temperature was controlled, the humidity was not reduced low enough to prevent condensation. In another case gold plating on transistor leads when the equipment was placed in long term storage caused stress corrosion cracking in large numbers of transistors. These problems are typical of those encountered in Air Force electronic equipment.

Bonded structures are also susceptible to corrosion attack at the bond line, generally by moisture penetration at exposed edges. At present no primary aircraft structure relies solely on bonding but there is a large amount of honeycomb sandwich (bonded) structure in most modern aircraft. It appears as fairings, control surfaces, floors, radomes etc. Significant failures of such structures have occurred due to bonding failures particularly at edges where load is transferred into other structure. When one face becomes separated to any extent the bending and shear capability can be significantly reduced.

The actual strength reduction due to corrosion effects in structure is not well known. Often corrosion interacts with fatigue, and corrosive pits act as stress concentrators and fatigue crack initiation points. Thus for a corroded element there would generally be a reduction in strength due to net material section decrease as well as due to a fatigue crack. For electronic equipment, the equipment will be more susceptible to normal in-flight vibration (which is of course not a hardness consideration) as well as the major shock and vibration levels due to a nuclear encounter. Normally however, electronic equipment is not shock tested during its service life, and it is very doubtful that normal electrical functional tests would discover an increased susceptibility to shock and vibration damage due to corrosion.

2.3 SKIN REFLECTIVITY

The amount of thermal energy deposited in a structural skin depends on the surface reflectivity. Highly reflective surfaces can significantly reduce the

vulnerability of thin skin sections to damage by thermal heating. The highest reflectivity that can be achieved by "reasonable" procedures currently is for freshly painted white surfaces using titanuim dioxide pigments. Using a high proportion of pigment and minimum binders it is possible to achieve a reflectivity of about 85% for nuclear thermal radiation. Normal weathering and aging soon reduce the initial high reflectivity of such paints so that after a few months or a year reflectivity values of 65% to 75% would be expected. Reflectivity continues to deteriorate with age, paints eventually lose adhesion to the substrate and tend to become brittle or chalky and easily eroded. During the early years of U.S. strategic bomber forces there was emphasis on delivery of large yield weapons and the necessity of maintaining high reflective paint on the aircraft surfaces exposed to thermal radiation during safe delivery maneuvers. Reflectivity of white painted surfaces was measured periodically and it was not unusual to repaint every eighteen months or so.

In recent years the durability of paint binders has greatly increased, particularly with the utilization of polyurethane or silicone as a dispersing and binding material. These paints were developed primarily for corrosion protection of exterior aircraft surfaces and Military Specifications now require that all military aircraft have a 3 mil (or more) coating of corrosion resistant paint on all exterior surfaces. Pigment color may be as required for identification or camouflage or for thermal reflectivity. Under normal weathering and aging conditions corrosion resistance paints are adequate for five years or more for corrosion purposes. Adhesion to the substrate is excellent and it has become common practice to simply add new coats of paint over the old paint whenever new paint is needed rather than stripping back to bare metal or composite surface. Under normal painting procedures each coat of paint adds about one mil of thickness. Up to twenty mils of total paint thickness can be accumulated before mechanical problems of paint checking, curling, or debonding from the substrate force stripping back to the substrate. In some cases the extra weight of added coats of paint may be unacceptable so stripping is done to reduce weight before repainting.

The amount and color of pigment that can be added to corrosion resistant paint can vary widely so the reflectivity currently available with MIL SPEC acceptable paints can vary widely. In a series of tests conducted at Air Force Materials

Laboratory in 1974 through 1975 (Ref. 4) the measured absorption coefficients for a specified thermal pulse for some selected paints were as follows:

PAINT	THERMAL ABSORPTION COEFFICIENT	
	Over 32 mil aluminum substrate	Over 40 mil or 100 mil fiberglass composite substrate
White Polyurethane	.132	.173
White Silicone	.191	.193
Aluminized Polyurethane	.239	--
Aluminized Silicone	--	.324
Gray Polyurethane (36622)	.440	--
Gray Silicone	.491	.499
Olive Drab Silicone	.667	.673
Olive Drab Polyurethane	.885	--

2.4 EQUIPMENT AGING AND SHOCK DAMAGE

Subsystems on the aircraft such as hydraulics, avionics, landing gear, pneumatics etc. are generally not susceptible to direct damage from the blast, overpressure and thermal effects from a nuclear burst. However they may be damaged by the shock and vibration due to the blast impinging on the aircraft structure. In particular electrical connections may be dislodged and equipment supports may be broken or bent so that subsystems can no longer function. For the aircraft in a delivered condition, meeting all hardening and performance specifications, these failures are remote but aging can induce brittleness, attachment looseness, and general weakness.

Very little systematic work has been done to determine the shock and vibration resistance of aged avionics equipment. There has been a program of hardness surveillance for the Minuteman missile system and in general no significant degradation has been observed in the five years of surveillance. Two areas of possible concern have been identified however. The first relates to the degradation of elastomers used in shock mounting of equipment. Ozone can cause

significant embrittlement of typical elastomers over a long period, and although this is not a problem in the silo environment, it could be significant in the aircraft environment, with high altitude exposure where the ozone concentration is increased. The other concern relates to connectors. Frequently, with removal and replacement of equipment a connector pin may be misaligned and forced back. The electrical connection is made, but the pin is dislodged in the connector, and a shock could break the connector.

It also appears likely that in an aircraft environment with vibration generally present, mechanical and soldered joints could become weakened over a long period (say, 10 years) to the point where a major shock could cause failure.

Radomes may also be subject to strength degradation due to debonding of the honeycomb from which many are made. Corrosion has been mentioned as a damage mechanism for honeycomb structures, and impact damage can also cause debonding and core damage. Impacts in flight are generally caused by hail and bird strikes, and major damage to radomes has been caused by both of these. However, it is not the obvious, major damage that is of concern here, but the steady, unnoticed damage caused by light to moderate encounters with hail (or birds). Such damage, where the core is slightly crushed and the core/skin bond damaged could accumulate so that the blast and overpressure resistance of the radome was degraded and the radome could be destroyed at less than its specified hardness level. No data is known to be available on residual strength of radomes to blast and shock in a damaged condition.

3.0 CRITICAL ELEMENTS AND FAILURE MODES

In a nuclear burst encounter different elements of the aircraft are "critical" for a given hardness level depending on the weapon size, the burst orientation with respect to the aircraft and the flight conditions, i.e. velocity, altitude, and gross weight. These elements must be listed, their failure modes determined, and then the failure critical mode compared with the degradation mechanisms. If there is a possibility of hardness degradation for a critical element, then this element must be studied further.

3.1 STRUCTURE

Previous work on mission completion (Ref. 5) gave a procedure to be followed to determine critical elements or components and their damage modes.

For the base escape situation of a wide body commercial cargo jet, a preliminary screening analysis led to the identification of 16 elements/components of the aircraft for further detailed analysis.

a. Elevators
b. Nose radome
c. Ailerons
d. Horizontal stabilizer
e. Stabilizer panels
f. Windshield
g. Wing
h. Spoilers
i. Crown skin
j. Side body panels
k. Vertical fin
l. Fin panels
m. Rudder
n. Fuselage skin at section 47
o. Fuselage frame at section 46
p. Wing leading edge

The sequence of failure of these elements was then determined along each of the burst orientation rays. For an overhead burst, one such sequence was:

Element	Damaging Effect
Stabilizer	Gust
Elevator	Thermal/Overpressure
Radome	Thermal/Overpressure
Aileron	Thermal/Overpressure
Stabilizer panel	Overpressure
Windshield	Thermal
Wing	Gust
Spoiler	Thermal/Overpressure
Crown skin	Overpressure
Frame	Overpressure

The failure modes can be summarized by groups as follows:

Windshield

Thermal The front cockpit window delaminates and becomes opaque. Heat from the nuclear flash and fireball is absorbed in the plastic and plastic/glass bond. Older windshields suffer discoloration from U-V rays and can absorb more thermal radiation, to cause damage at a greater range.

Elevator
Aileron
Spoiler
Radome

Thermal/Overpressure These components are of honeycomb sandwich construction. The front face delaminates due to heating of the thin skin by the thermal pulse, and the subsequent shock and overpressure crush the honeycomb and buckle the component. Subsequent flight loads can completely collapse or remove the structure. These components can be weakened by corrosion (debonding) or made more susceptible to thermal damage by surface reflectivity degradation.

Crown Skin

Overpressure Fuselage and stabilizer panels deform

Frames Stabilizer Panels	inwards between supporting members such as stringers, frames or ribs. At greater overpressures the upper fuselage surface is crushed inwards to form a "dent". In this case the frames are cracked and sheared. Corrosion and fatigue cracks can weaken the basic structure to reduce its ultimate strength.
Wings Stabilizer	Gust These members respond dynamically and break off at about the half span region. Analysis shows that there are several cycles where the peak loads are roughly comparable. Whether the upper surface fails in tension or the lower surface in compression was not established. Fatigue cracks would be the major cause of a reduction in ultimate strength for a tension failure.

3.2 EQUIPMENT

All internal equipment is originally qualified to some specified shock and vibration spectrum, but seldom is this related to the total vehicle response to a nuclear blast. Generally equipment is mounted without regard to the location of structural nodes or anti-nodes in the total airplane dynamic motion. However it appears very reasonable that a gust of sufficient strength to nearly fail the wings or tail could impose dynamic loads on equipment mounts, fittings and connections sufficient to fail them. To the best of our knowledge, no systematic study has been made of the equipment most susceptible to such shock and to vibration failure. Thus "critical elements" cannot at this stage be defined, except to indicate that most flight control navigation and weapon delivery avionics boxes are mission critical, and hydraulic systems are also critical.

The failure mode of all of these items is similar, that is connections (electrical or mechanical), support brackets, or fasteners fail, probably in tension, due to violent shakings. Any corrosion, cracking, joint sloppiness or shock mounting hardening, would increase the probability of failure of such components.

4.0 MAINTENANCE PROCEDURES

For convenience in examining the implications for fleetwide hardness variation maintenance procedures may be considered in four categories.

1. Routine maintenance which does not affect configuration and does not include disassembly of items.
2. Periodic scheduled maintenance which includes teardown, inspection, repair or refurbishment, and reassembly of specified items.
3. Unscheduled fleetwide inspection, repair or refurbishment or replacement of items discovered to be deficient.
4. Unscheduled repair or replacement of items on single aircraft due to accidents, excessive wear, parts failure or unusual events.

Category 1, routine maitenance, is expected to have very little influence on hardness variations among individual aircraft. This type of maintenance includes replacing worn tires, spark plugs replacement, oleo pressure adjustments, servicing of air, oil, fuel, hydraulic systems, batteries, oxygen, and fire suppressant systems and external visual inspections of engines, parts, drains and all external surfaces. The only significant effect on nuclear hardness anticipated from variations in this category of maintenance is in the cleanliness of exterior surfaces and resultant changes in skin reflectivity. Oil leaks, smoky engine starts, and ground operations on dirt fields are the most frequent causes of dirty exterior surfaces. The maintenance work load, weather conditions, and personal zeal of commanders and maintenance personnel determine how frequently aircraft are cleaned and how carefully all traces of dirt are removed. There are no flight safety requirements to dictate that a given amount or shade of soiled skin must be corrected so there is expected to be considerable variation in amount and degree of surface grime that is tolerated.

Category 2, or scheduled teardown maintenance and inspections, normally covers items that have a definite wear rate and are necessary for flight safety or for mission accomplishment. Engine overhauls are a typical example. Scheduled inspections for fatigue cracks in structural members are another example. Items

with friction wear - such as engines, generators and pumps seldom directly influence the nuclear hardness of an aircraft. Items inspected for fatigue or aging degradation however have a direct relationship to hardness. Inspection intervals are selected to provide high assurance that deterioration will be found well before flight safety is affected. Repairs or replacements are not usually implemented though until some specified amount of deterioration has accumulated. The amount of deterioration depends on flying hours and severity of flight loads in addition to random material properties. Repair or replacement of deteriorating items are planned on a schedule that will keep the fleet above the specified tolerance values assuming normal operating conditions and normal flight loads. The schedules for individual aircraft refurbishment can be adjusted if inspections show faster deterioration than expected. This results in shifting flying duties to aircraft with greater safety margin and consequently faster deterioration of those aircraft. Tight budgets further dictate that repair and replacement of deteriorating items be kept at the lowest level consistant with the specified minimum safety margin. The fleet wide distribution of deterioration therefore tends to be closer to the minimum acceptable values than to the best practical achievable values. The condition of individual aircraft, such as number of fatigue cracks, condition of the paint surface, number of loose or failed rivets and dents or bulges indicating severe air loads are usually well documented by the scheduled maintenance crews so that surveys of maintenance records can establish the actual distribution. The effects on nuclear hardness deterioration can be calculated and applied to the distribution determined from maintenance/inspection records.

Category 3 or unscheduled inspections or repairs are the results of unanticipated defects. They are triggered usually by the unexpected finding of a serious structural defect or weakening such as long fatigue cracks in a structure not usually associated with fatigue cracking. Sometimes an in-flight accident leads to discovery of a serious defect. There have been numerous instances where discovery of unexpected defects have led to grounding a fleet until repairs can be made or inspections completed to assure that continued flight is acceptable. Generalizations about the likelihood of undiscovered defects throughout a fleet are difficult. Past experience indicates that the introduction of new structural materials nearly always introduces new problems. Changes in operating modes, such as switching from high altitude to low altitude

flight, also nearly always introduce new failure modes or accelerate deterioration modes that formerly were unimportant. The longer a fleet has been operating and the more thoroughly all potential operating modes have been exercised the fewer unexpected defects remain. The fleet may be getting older, more fatigued and have lower safety margins but there are fewer surprises. Unscheduled inspections to determine the extent of fatigue cracks in a previously unexpected structure are replaced by routine scheduled inspections that monitor the development of fatigue cracks and provide repair or replacement on a timely basis.

Unscheduled fleetwide inspections for potential structural deficiencies should give a good measure of the actual distribution across the fleet since such inspections are usually high priority and are conducted uniformly on all aircraft within a short time. Reviewing such records will determine the distribution of specific defects at the time of the inspection. There are only a limited number of cases to draw upon however. While there have been numerous cases of fleet grounding to determine whether potential defects existed, the large majority of such cases end with negative results. These few cases can give valuable information on the distribution of selected defects. In inspections for fatigue cracks the results have been consistant with laboratory developed models of the rate of crack growth as an exponential function of load cycles. Inspections of paint surfaces have shown that paint durability is strongly influenced by priming and curing conditions. Improper priming, drying times and curing conditions can lead to fragile paint surfaces that will be stripped away by the first flight through heavy rain. For a given quality of paint deterioration is almost linearly related to severity of weathering so if proper quality paint exists initially, then the distribution of deteriorated paint surface is nearly linear with time since repainting.

Category 4, or unscheduled repair of individual aircraft occur frequently as the result of observed equipment failures in-flight. Most of the failures are burn out of electrical components with mechanical failures of motors, pumps or switches much less frequent. In flight overheating and smoking of electrical circuits with possible charring of adjacent wiring insulation sometimes occur. Ground accidents by contact with refueling trucks, tow trucks, maintenance scaffolds or bomb dollies cause dents, scratches, chipped paint or bent

antennas. Serious accidents are visible and soon repaired but the human tendency to hide minor accidents leads to an accumulation of minor scratches and dents, sometimes covered by paint. This can result in increased vulnerability to corrosion near the damaged areas. The importance of corrosion in inducing electrical contact failures and reducing the effectiveness of seals, fittings and closures is amply demonstrated in practice but direct corrosion effects on reducing the strength of structural members have not appeared as significant problems in operational aircraft. With the continued aging of existing military aircraft the potential effects of long term corrosion on structure may become more important. The anticipated deterioration of mounting gaskets, sealing compounds and insulation is expected to lead to an increasing number of vibration induced mechanical failures of electrical equipment. This would result also in mechanical damage to a greater number of electrical items by a nuclear blast encounter. Monitoring the number of equipment failures over the years would lead to some information on increasing fragility with age and increasing probability of mechanical shock damage to equipment by blast encounters.

5.0 METHODS FOR ESTIMATING HARDNESS VARIATION

Previous sections have identified physical processes by which aircraft components and subsystem lose their as-built strength and become susceptible to damage at reduced hardness levels. This section suggests methods for obtaining a quantitative measure of the variation of this actual hardness in an aircraft fleet.

5.1 CRACKS

For safety of flight consideration, the existence and growth of cracks anywhere in load bearing structures is of major concern. However, for hardness degradation the focus of attention is more limited, since the only members that need be considered are those which are critically loaded in tension as the aircraft responds to a nuclear burst. This generally reduces to a particular section of the wing, body. fin or horizontal stabilizer which reaches its design ultimate load in a gust response mode at the critical overpressure level. Thus if the population of cracks at a critical structural section can be determined for an aircraft fleet the corresponding distribution of the strength (or hardness) degradation can be inferred. Fortunately, due to the great concern for structural integrity of aging aircraft (both civil and military) over the last few years, there is a good and growing knowledge of crack populations.

Figure 6 gives a schematic flow of the tasks which must be accomplished, and Figure 7 indicates the form of the inputs used to make the estimate of hardness variation. Each of the items in figure 6 is discussed in detail below.

5.1.1 Critical Hardness Areas (Tension)

The critical areas of the aircraft can be determined by use of a suitable gust response computer code such as Vehicle Inelastic Bending Response Analysis (Vibra) 4, Vibra 6 or some other dynamic response code with which the analyst is familiar. The solutions will give these sections of the wing (or fin or stabilizer) where the dynamic load just equals the ultimate bending moment capability of the structure. Different burst orientations must be considered,

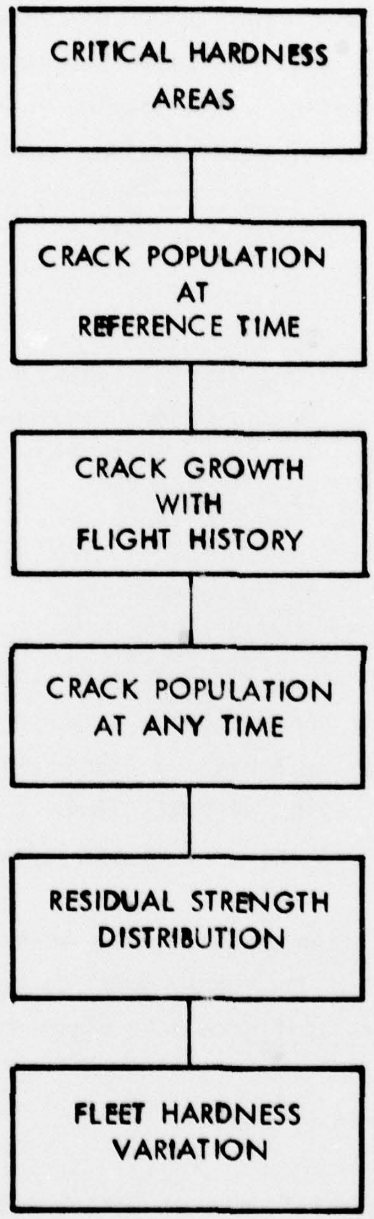

Figure 6 Task flow to determine hardness variation due to cracks

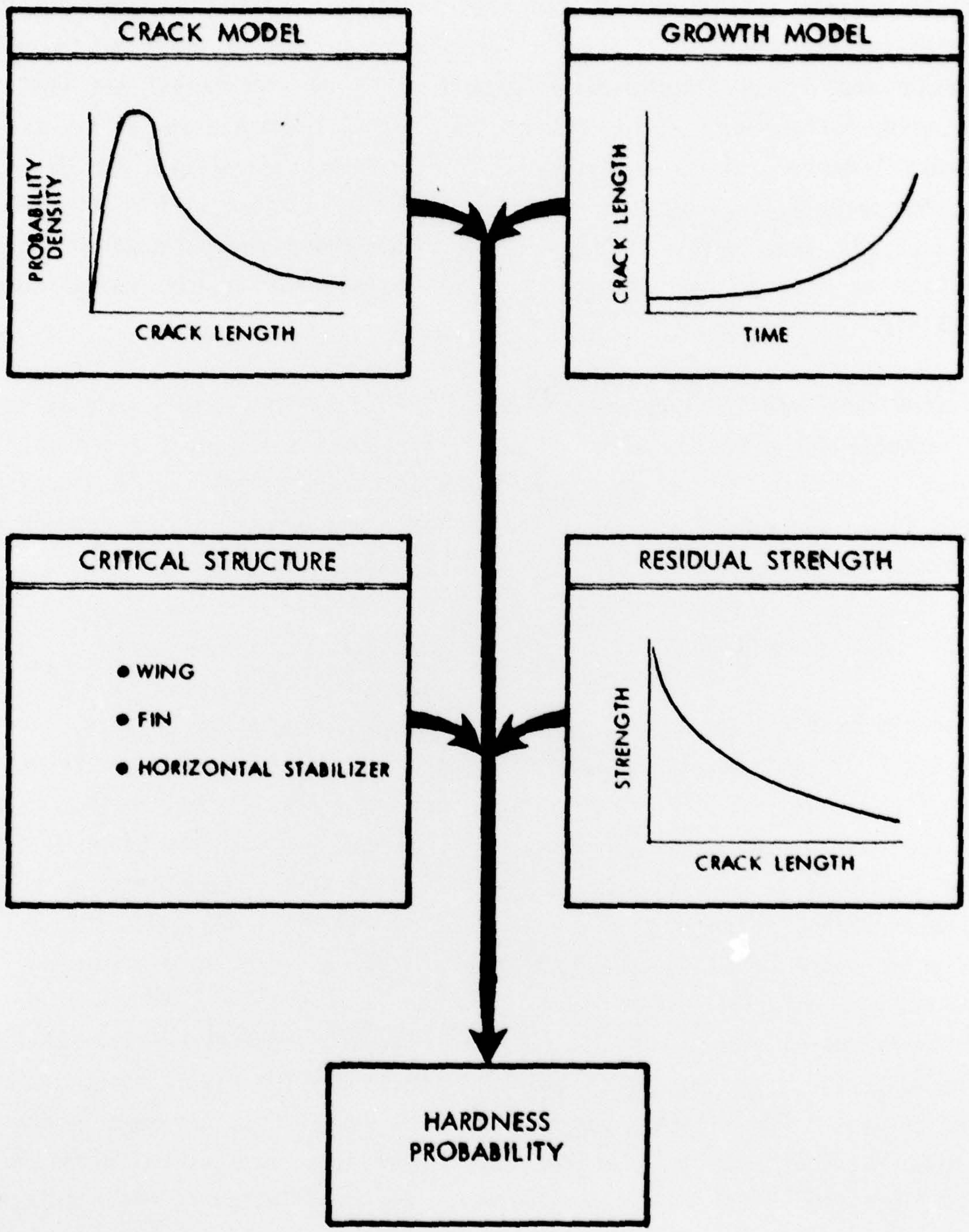

Figure 7 Basic hardness variation inputs

of course, but generally, due to the basic dynamics of the structure, the critical regions are located at the same sections. Both the lower and upper surfaces can be critical in tension for the burst may be above or below the aircraft and a well designed structure will be "balanced" so that the compression surface will yield at about the same ultimate loading as the tension surface. However, it is necessary to verify this capability, and to ensure that, for example, in a burst from above, the upper surface will first yield in tension. In some cases it may have been designed for some other loading condition so that to take a maneuver compression load it has excess tension capability.

One other consideration should be noted. In several Vibra-4 runs made to verify the response of a KC-135 wing, it was found that the second response cycle imposed loads that were as severe as the first peak. That is, if significant damage (i.e. ultimate load) is caused by the first peak load, the second will cause destruction.

5.1.2 Crack Population

The cracks existing in a fleet of aircraft cannot realistically be obtained by individual inspection. Instead, the method used has been to make a very exhaustive survey of one wing or one horizontal stabilizer during a major tear-down maintenance overhaul and use the cracks thus observed to project to a fleet distribution. Based on theoretical statistical considerations, a 2-parameter Weibull distribution can be used to realistically represent a fleet population based on a average crack size and the extreme crack observed. For most current aircraft fleets such distributions have been obtained at some relatively recent time. For civil aircraft fleets, manufacturers are obtaining individual high time components and performing detailed examinations including rivet removal. For military fleets there are some total tear-down inspections on older aircraft, as well as detailed inspections of isolated areas on the total fleet (eg. C-130 wing access holes). For newer aircraft the requirements of the structural integrity program should generate a detailed inspection for cracks on high time aircraft.

It is important to observe that this crack population distribution for nuclear

hardness is only required at the critical sections identified in 5.1.1. It is conceivable that these sections are not those normally inspected for cracks. In this case unique inspections will be required for nuclear hardness. Further, the cracks of interest here generally are much smaller than those of concern to the structural integrity of the aircraft so care must be taken that the very small cracks are identified.

5.1.3 Crack Growth

The cracks existing in a structure at any given time will continue to grow at a rate depending on the applied loads. Currently there is increasing understanding of the rate of growth of cracks in loaded structures, and suitable inspection periods are being defined depending on the material, the built-up structure and the flight history, so that long cracks are observed and repaired long before unstable crack growth can occur. These crack growth predictions can be used to determine the crack population in the nuclear critical members at any given time in the fleet history. If a portion of the fleet is subject to different flight loads from the majority, then of course their crack growth must be computed separately.

5.1.4 Residual Strength

In section 2.1 (Tables 2 and 3) data was given relating crack length to a reduction in ultimate strength. Based on this data it is possible to convert the crack population density obtained in 5.1.3 to a reduction in strength probability density. However it must be noted that the quantitative basis for this step is very limited, and new tests should be performed relating the material of interest in a particular aircraft to its reduced strength. It would be extremely desirable also to test actual built up panels, as well as test coupons. Even with this proviso, however, it appears that if the average crack is of length 0.1" in the critical structure the ultimate strength will be reduced by about 5% - 10% for 2024 type material.

5.1.5 Fleet Hardness Variation

The last step is to convert from reduced strength to an equivalent nuclear

hardness level. This step must use the results of 5.1.1 which gives the details of the loading on the critical member of interest. In some cases this critical member will be lightly loaded in flight, so a reduction in ultimate strength means about the same proportional reduction in hardness. In other cases, the member will be highly loaded and only say, 50% (an extreme case) of its strength is available to take the nuclear load. In such a case the hardness is reduced much more than the first case. Thus the determination of hardness variation in the fleet must address each critical member in turn to obtain a realistic extimate. Detailed, highly accurate calculations are not necessary (and probably not obtainable) for a gross assessment of hardness variation, but some detailed gust response calculations are essential, with engineering judgement used to interpolate and extrapolate to other conditions.

5.2 CORROSION

As for cracks the first step to be taken to determine the effects of corrosion on hardness is to determine hardness critical members which are subject to corrosion attack. This could be accomplished by a review of maintenance reports for the aircraft fleet to determine what corrosion had been observed during regular maintenance inspections. It is highly unlikely that any primary structure would be only damaged by corrosion as opposed to corrosion assisted cracking. The possibility of a frame or stringer etc. being greatly corroded sufficiently to reduce its strength is remote, but it should be checked by a study of maintenance records.

The only type of structure that appears susceptible to a reduction in strength due to corrosion is bonded structures, primarily honeycomb secondary structures, but also in the future, possibly primary structure. The damage in this case is a failure of the bond and various non-destructive inspection techniques are available to determine the extent of debonding. If it is determined that a particular bonded structure is nuclear hardness critical, then an analysis must be made to determine the reduction in strength versus corrosion damage. Depending on the results of this analysis inspection requirements could be developed ranging from a detailed tear down of the component if it is highly strength sensitive to damage, to normal non-destructive inspections techniques for many damage sensitive areas. Based on the inspection of say, 2 or 3 of the

identified structure, an estimate might be made of the corrosion damage existing in the fleet, and thus of the hardness variation.

5.3 SKIN REFLECTIVITY

Since thermal damage is critical for thin skinned (generally honeycomb sandwich) structure, it is only necessary to consider these structures. Thus the comments below apply in general to control surfaces, radomes, and possibly fairings. Since thick skin sections (such as the wing box) are not critical for thermal damage, their reflectivity need not be considered.

Skin reflectivity directly effects the amount of thermal energy absorbed from the nuclear burst, and the resultant heating and damage of substrate skin material. Fiberglass has an absorption coefficient of about 0.9 for nuclear thermal radiation, while weathered aluminum is about 0.5 and fresh white polyurethane paint is about 0.13 to 0.17. This corresponds to a reflectivity of about 0.85 for fresh paint, which decreases by weathering and surface grime to about 0.6. In a typical fleet of military aircraft the paint scheme is not uniform for all aircraft, some aircraft may be camouflaged, and the freshness of the paint will vary depending on the time from the last paint job. In the past, this decision on when to repaint was usually a subjective judgement based mainly on visual appearance and partly on the need for corrosion protection. No data base of reflectivity measurements exists in the maintenance records from which a distribution of relfectivity variation can be made.

Accurate measurements of skin reflectivity require laboratory measurements. No light sources are available to accurately simulate the thermal spectrum from a nuclear detonation so that spectral measurements should be made to permit the comparison of the laboratory source with the desired nuclear burst. More than one nuclear pulse is possible of course, and the equivalent black body temperature of a nuclear fireball will depend on height of burst, atmospheric conditions, and weapon yield. This can cause the reflectivity of the same surface to vary from 10% or more from one burst condition to another. If the spectral distribution of probable threat bursts is known, then reflectivity for each burst could be determined from one set of laboratory measurements. Correlations with other light sources could also be made in the laboratory so

that portable light sources and reflectometers could be used to measure the reflectivity of individual aircraft. Such equipment is relatively expensive and demands considerable operator skill.

A much more simple technique would be to assume that degradation in reflectivity is directly related to the weather exposure, time and total flying hours. A linear distribution could be used between the best, freshly painted surface (reflectivity 0.85) and a deteriorated surface ready for repainting (reflectivity 0.6). A suitable time period can be determined from maintenance records for the surface finish to be ready for repainting and then the fleet distribution could be determined.

To finally obtain the variation in thermal hardness, analyses would have to be made using Thermal Response Analysis Program (TRAP) or some other suitable program, to determine the actual hardness reduction of the critical surfaces due to the increased absorption of thermal radiation. Since the increase is of the order of three times the hardest surface (15% absorbed to 40% absorbed) there will be a significant reduction in hardness.

5.4 AGING EQUIPMENT

There is no data base available on the possible descrease in hardness of aircraft equipment with age. Failures of avionics boxes, or equipment supports are noted during operations or maintenance, but a decrease in capability to withstand originally specified shock and vibration is not measured. Further, this degradation does not lend itself to a theoretical prediction. In fact it is not clear what level of shock and vibration any particular equipment would experience in a nuclear burst. In some more recent systems the shock and vibration spectra imposed on the equipment has included some consideration of a nuclear burst, so that the specs are a convenient reference point. That is, it would be of considerable value in estimating the current hardness of any aircraft to determine how well the equipment meets the original shock and vibration specifications.

Tests to determine this capability could be readily made during routine major maintenance, when the aircraft is out of service for a significant time.

Selected avionics boxes or circuit cards would be chosen, based on their mission critical nature and tested for shock and vibration capability. Details of the actual test would depend on engineering judgement and analysis of the expected environment, but the simplest test would be to subject the box to the specification levels of shock and vibration. If it still meets these levels after 5 or 10 years in service, then all is well. If some of the boxes fail their functional test after this exposure, then more detailed tests would have to be made to determine their actual capability, and to relate this shock resistance to a nuclear hardness level.

6.0 CONCLUSIONS

A methodology has been developed to determine the variation in nuclear hardness in a fleet of aircraft. The actual reduction in hardness could be calculated for a given aircraft by using existing computer programs and identified critical members or areas of the aircraft.

It appears reasonable to estimate the variation could be of the order of 20% for both gust and thermal response. The accuracy of the calculations would be of the same order as the accuracy of basic hardness assessment calculations.

Potential damage mechanisms included cracks, structural corrosion, paint deterioration, and aging of equipment, particularly avionics.

7.0 RECOMMENDATIONS

1. A sample calculation should be made for reduction in hardness due to cracks for one critical member of a suitable aircraft.

2. Residual strength vs. crack length tests should be made for specific materials and structural members appropriate to aircraft of interest.

3. For thermally critical areas the reflectivity should be measured on several strategic aircraft to cover the range of freshly painted to well weathered.

4. A cost study should be made to determine the additional costs incurred as a function of improved reflectivity i.e. increased frequency of painting.

5. Sample older avionics black boxes should be shock tested to determine their capability to meet original specifications.

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